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LOW-THRUST TRAJECTORY BACON PLOTS FOR MARS MISSION DESIGN

Ryan C. Woolley[*], Frank Laipert[†], Austin K. Nicholas[‡], and Zubin Olikara[§]

The best way to understand a mission design trade space is by creating a good map of all the possibilities, and by knowing how to read it. Bacon plots, which are low-thrust analogs to porkchop plots, give insight into key parameters and sensitivities of possible transfers to Mars (or other destinations). They are mission and parameter specific, and can be created to represent single or multiple legs. This paper outlines the creation process and lessons learned in interpreting the results. An example is given on how bacon plots have been employed for complex mission-level optimization.

INTRODUCTION

The optimization of low-thrust trajectories is a difficult problem that requires simultaneous optimization of many design parameters. Under certain assumptions, such as near impulsive burns and constant-thrust multi-revolution spirals, analytic solutions exist. However, these solutions do not apply when traveling from Earth to most deep space destinations, such as Mars. Typically for these missions, trajectories are analyzed using computationally intense methods that find a locally optimized solution. Mission requirements and flight systems are then designed around this solution, which is generally not a globally optimized solution in terms of mission mass, cost, and durations. System parameters need to be traded alongside trajectory optimization in order to design the best mission possible. Useful tools and techniques are needed to aid in this process.

Traditional ballistic transfers to Mars are well characterized by performance maps known as porkchop plots. They show key parameters such as launch and arrival velocities in contours as a function of launch and arrival date. The contours allow the user to quickly identify the optimal trajectories that meet mission requirements. They also show what happens when dates are shifted or when certain constraints must be met. This method of portraying trajectory parameters as contours in the launch date/arrival date space is very useful for low-thrust missions as well. Recent papers have discussed the characteristics[1,2] and uses[3,4] for various types of bacon plots. Bacon plots differ from porkchop plots in that they are mission specific. That is to say that trajectories are not unique, as they are ballistically. An appropriate figure-of-merit, as well as the characteristics of the propulsion system, power system, and masses, must be taken into account in order to optimize

[*] Mission Design Engineer, Inner Planets Mission Analysis, Jet Propulsion Laboratory, California Institute of Technology, 4800 Oak Grove Drive, Pasadena, CA 91109.
[†] Mission Design Engineer, Flight Path Control, Jet Propulsion Laboratory, California Institute of Technology, 4800 Oak Grove Drive, Pasadena, CA 91109.
[‡] Systems Engineer, Project Systems Engineering and Formulation, Jet Propulsion Laboratory, California Institute of Technology, 4800 Oak Grove Drive, Pasadena, CA 91109.
[§] Mission Design Engineer, Inner Planets Mission Analysis, Jet Propulsion Laboratory, California Institute of Technology, 4800 Oak Grove Drive, Pasadena, CA 91109.

the trajectory for each launch/arrival date pair. In order to consider, for example, multiple power levels or multiple engine choices, many bacon plots must be created. In this paper, we discuss some of the considerations that go into bacon plot creation. We also discuss trends, lessons learned, and applications for their use in mission design.

Solar electric propulsion (SEP) technologies have come a long way over the past decade. While Russian and other foreign satellites have been using electric propulsion for station keeping since the 1960's, it was not until more recently that the commercial US market started regularly adopting the technology[5]. Now it is regularly used in commercial spacecraft buses, both for stationkeeping and orbit raising. By 2017, over 248 spacecraft had employed electric propulsion in Earth orbit.[6] The vast majority are either gridded ion or Hall-effect thrusters, which are technologies replacing the arcjet systems of previous decades. SEP has also been used to go beyond Earth orbit, reaching destinations such as comets (Deep Space 1, 1998)[7], asteroids (Hayabusa, 2003; Hayabusa2, 2014)[8,9], the moon (SMART-1, 2003)[10], to the protoplanets Vesta and Ceres (Dawn, 2007)[11], and Mercury (Bepi-Colombo, 2018)[12]. SEP missions are currently being planned or proposed to go to a metal asteroid (Psyche, 2022), return samples from a comet (CAESAR, CONDOR), deflect asteroids (DART), and return samples from Mars (MSR), amongst many others.

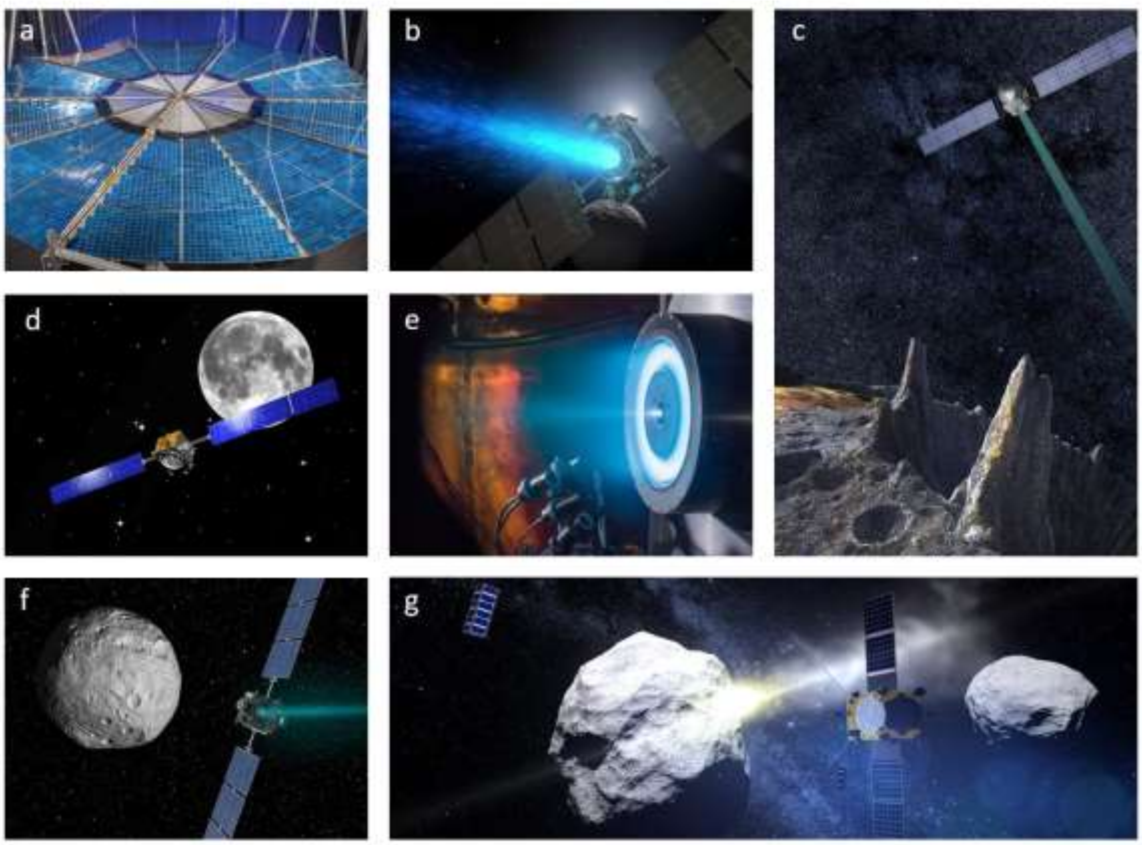

Figure 1 - SEP missions and technologies. a) 10-meter (20 kW) MegaFlex solar array, b) Dawn approaching Ceres (2015), c) Psyche mission (concept, 2022), d) SMART-1 (2003) to the moon, e) HERMeS 12.5 kW Hall thruster, f) Dawn approaching Vesta (2011), g) Double Asteroid Redirection Test (DART, concept).

The allure of solar electric propulsion is that it allows deep-space missions to carry more cargo and use smaller launch vehicles while reducing mission costs. It provides such high fuel economy, in the form of high specific impulse (I_{sp}), that it reduces the amount of propellant required for these missions by as much as 90 percent. In addition, the near-constant thrusting of the engines opens the possibility of trajectories that are not achievable by traditional propellants. For many missions, the use of SEP is not only beneficial, but enabling.

The trade-off for such high I_{sp} is two-fold: first, large solar arrays are necessary to provide sufficient power, and second, the thrust provided is miniscule in comparison to traditional thrusters. Currently, there are many investments to develop solar arrays that are lighter, stronger, more compact, and less expensive than those currently available. These arrays help offset the mass and cost penalties associated with the need for high power, which is often in the 10's of kilowatts. There are also many high-power (> 4 kW) engines both existing and in development. These include NEXT (NASA/Aerojet), XIPS (L-3/Boeing), XR-5 (Aerojet)[13], SPT-140 (SSL/Fakel)[14], PPS5000 (Safran)[15], T6 (Qinetiq)[16], along with engines greater than 10 kW being designed and tested.[17]

There exist a multitude of low-thrust trajectory optimization techniques and tools – ranging from simple scripts to high fidelity, commercial, flight-proven software. Since bacon plots are typically useful in the early formulation stage, speed and flexibility are key. Any software that gives quick, reliable results would be suitable to make bacon plots. An additional bonus would be the option to parametrically sweep parameters while using nearby "converged" solutions as seeds. For our work, mission design analysis was carried out using MALTO (Mission Analysis Low-Thrust Optimization), a fast, medium-fidelity low-thrust optimizer developed at JPL.[18] Similar software such as Goddard's Evolutionary Mission Trajectory Generator (EMTG)[19] and the European Space Agency's DITAN (Direct Interplanetary Trajectory Analysis)[20] would be suitable as well. Higher fidelity tools, such as MYSTIC, GMAT, or Copernicus, could then be used to verify selected individual trajectories.

METHODS AND ASSUMPTIONS

In the early stages of Mars mission formulation it is difficult to know what parameters to select for an electric propulsion system. In fact, in some cases, it is not always apparent as to whether SEP will be the right choice or not. In general, missions that require more total ΔV stand to benefit the most. Typical Mars missions, in order of increasing ΔV, would include: direct entry landers, Mars fly-by, high Mars orbit (HMO), low Mars orbit (LMO), Mars fly-by with Earth return (e.g. cyclers), and Mars orbit with Earth return (e.g. sample return). All of these could need more ΔV by starting (or ending) in Earth or cislunar orbit. Figure 2 can be used to estimate the total ΔV required for various missions, depending on the desired start and destination. Note that the ΔV calculations for impulsive transfers is significantly less than their low-thrust counterparts. The higher ΔV's are more than offset by the superior I_{sp} of SEP, which is typically 5-10 times as much as chemical thrusters. In addition to saving propellant for higher ΔV's, missions that require unique timeline constraints, geometries, inclinations or asymptotes, or multiple orbits stand to benefit from (or require) SEP as well.

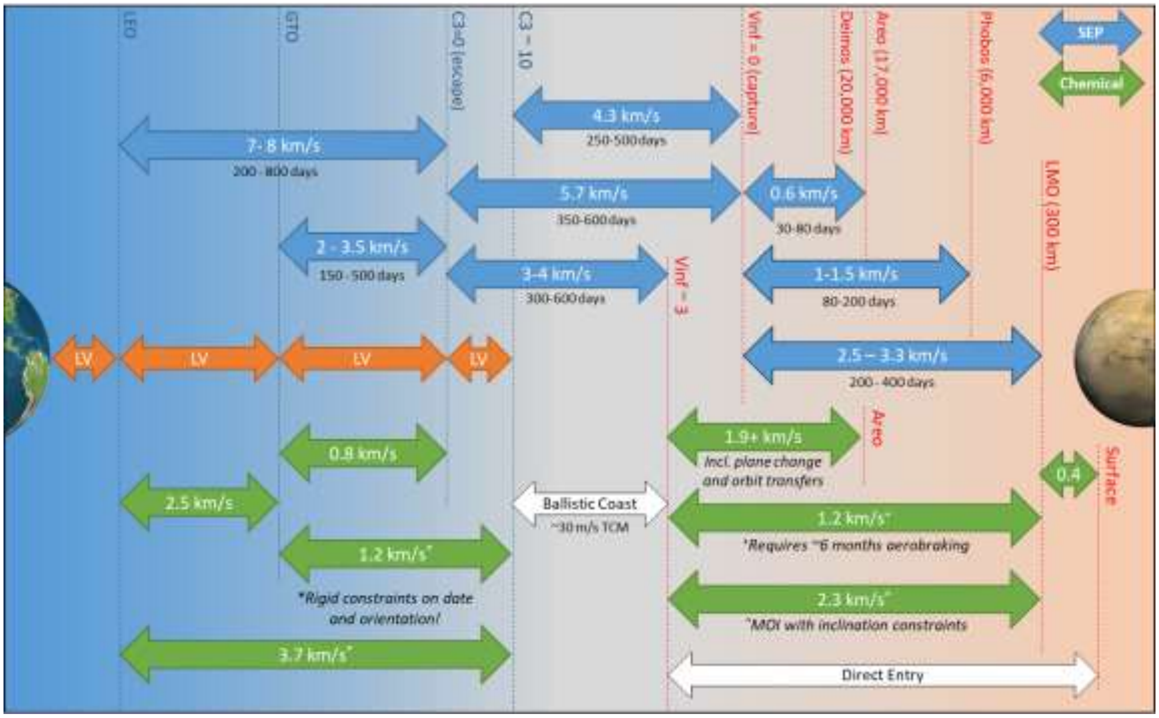

Figure 2 – ΔV requirements for Earth-Mars transfers using SEP (blue) or chemical (green) propulsion. Values are approximate and require mission specifics. Total mission ΔV can be estimated by adding arrows from starting orbit to final destination. SEP transfers typically require much more ΔV and time, whereas chemical transfers have stronger restrictions on dates and asymptotes.

It is difficult to discuss the optimization of low-thrust trajectories without knowing the detailed properties of the engines being used. To first order, it may be sufficient to assume a constant Isp and efficiency*. This is particularly true when the power level remains nearly constant over the trajectory, such as during spirals at Earth or Mars (the Martian eccentricity, however, does cause a fair bit of variation). In this case, the thrust is linearly proportional to input power. However, in reality, most thrusters vary in Isp and efficiency as power varies. In fact, the power-processing unit (PPU) can often vary both current and voltage to create multiple operating throttle points. Trajectory optimization routines can use these points, or a polynomial fit of thrust vs. power and mass flow rate vs. power, to optimize heliocentric trajectories that have varying solar power.

Another thruster parameter crucial to interplanetary SEP missions is the maximum throughput of each engine. This can be expressed in kilograms, hours, or total impulse. It is necessary to have thrusters that can provide the high ΔV's required for the trip to (and from) Mars – as much as 14 km/s or more, where more than half of the wet mass could be propellant. The key thruster parameters needed are: maximum and minimum input power [kW], thruster mass (with PPU, gimbal, etc.) [kg], thrust vs. power curve [N], mass flow rate (or Isp) vs. power curve [g/s or sec], and maximum throughput [kg, hrs, or Ns]. It is important to select thrusters that are sufficiently sized for the mass and ΔV of the desired mission. We have found that a good rule-of-thumb is to select engines and power levels to give ***initial acceleration levels of 0.15 – 0.3 mm/s²***. This is provided

* Thruster efficiency is defined as the ratio of the kinetic energy of the exhaust particles (or jet energy), to the electrical input power to the propulsion system. Typical efficiency values range from 40 – 65%. Peak efficiency is usually attained near the maximum input power of the thruster, and decreases as power drops.

by thrusters producing 150 – 300 mN per 1000 kg of spacecraft. As mentioned, it is also critical to use engines with enough throughput capability. It is possible to carry extra "spare" engines to cover the requirement, but the mass penalty can be high.

In addition to the appropriate thruster, it is also essential that they be adequately powered. The actual optimal power level cannot be determined until a full mission-modeling tool is in place. For missions that remain near 1 AU, or use a radioisotope power system, it would be practical to start with a system that fully powers the thrusters and spacecraft, with adequate margins. For solar missions to Mars, however, the input power can be decreased by 60% or more as solar distance increases. It is tempting to select a power level that keeps the engines fully powered throughout the mission, but this would mean much power is thrown away near Earth. This is where a tool that can estimate the propellant cost to carry additional power and determine the sweet spot is essential. There is also a trade between increasing power and decreasing time-of-flight (TOF). A good guess for a starting power level is to keep the thruster 60-80% of max power at Mars. Another way to get a rough estimate, which accounts for typical thruster performance, is to start with about ***5-10 W per kg of estimated wet mass***. It is acceptable to use even lower powers (at the cost of increased time-of-flight), but keep in mind the minimum power level to keep the engine above its lowest throttle point. In general, lightweight flexible arrays will provide the desired power level and mass savings. A reasonable starting target would be in the range of 7 – 10 kg/kW.

Once the basic mission profile, requirements, constraints, and possible thruster(s) are chosen, the next step is to determine how to explore the mission design trade space. The trade space of all possible trajectories can become impossibly large when considering the combinatorics of the input parameters. For example, evaluating all trajectories over a period of 3 years, with 4 thruster choices, 1-4 thrusters active, 10 power levels, and 3 launch vehicles could yield over 1.1 billion runs – obviously not a tractable number for early mission formulation. The desire here, however, is to sufficiently characterize the topography of the trade space with just a few strategic simulations that will give us the means to hone in on an optimal mission.

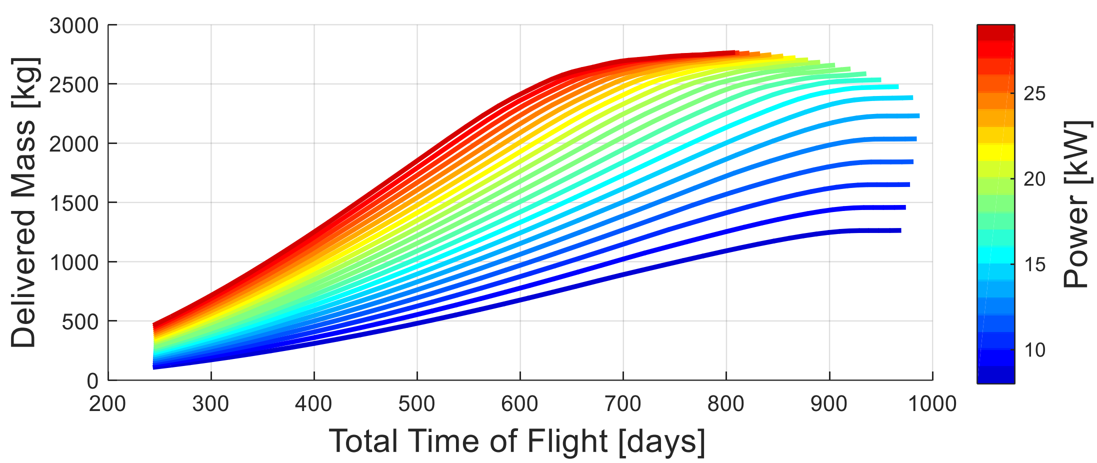

Figure 3 - Maximum delivered mass to LMO vs. TOF and power level. The time-of-flight includes both the heliocentric transfer and the spiral to the final orbit.

The first step is to decide which parameters, when varied, will make a good basis for the trade space. A common feature of most SEP trajectories is that there is a trade between propellant and TOF – the longer the duration the more efficient the trajectory. The relationship is typically asymptotic, often exhibiting a "knee" in the curve that represents a good sweet spot. Since early questions to answer in mission formulation are typically "how long will (or should) it take?" and "how much power does it need?", sweeping TOF and power to create a plot like Figure 3 would be beneficial. This plot is created for one thruster set for the maximum mass delivered to LMO by a given launch vehicle. Similar plots could be created for other thrusters or number of thrusters and easily compared. The data can be compared visually to give an idea of where to look for optimal conditions, or incorporated into a lookup database for a spacecraft design tool which iterates on system and trajectory parameters (see the section on Advanced Application for an example).

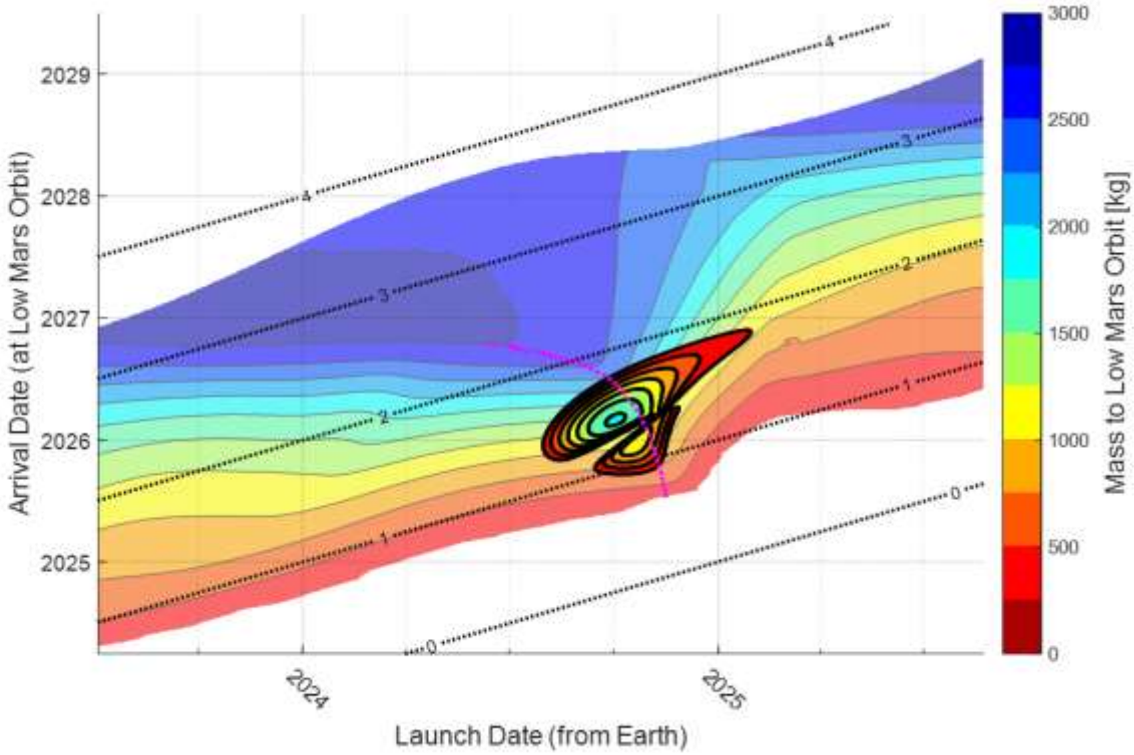

Figure 4 – Ballistic porkchop plot superimposed on a low thrust bacon plot. The launch date at Earth spans one synodic period (780 days). Diagonal lines show transfer times in years. Contour lines show the total delivered mass to Mars for a given launch vehicle – with blue being the highest. SEP allows for nearly continuous launch periods and increased delivery mass for longer flight times. The magenta dots represent the maximum mass delivered for a given TOF.

The limitation of the TOF plot in Figure 3 is that launch and arrival dates are free parameters, outputted by the optimization procedure. Mission constraints, however, are often tied to specific dates. In order to fully explore the topography of the possibilities, it is necessary to sweep the full range of launch dates (LD) and arrival dates (AD), which is the basis of the bacon plot. The contours displayed in the LD/AD* space can be any number of trajectory parameters from the optimizer output, but typically the most useful is to show maximum delivered mass (when given a starting constraint) or minimum propellant mass (when given an end constraint). Figure 4 shows a typical Earth to Mars bacon plot with maximum mass contours. As the colored contours progress from red to blue, more mass can be delivered to Mars for a given launch vehicle and low-thrust propulsion system. The contours for a ballistic porkchop plot (with aerobraking) are shown

* The LD/TOF space is a simple transformation of the LD/AD bacon plot, which is especially good for displaying multiple opportunities without the "band" of contours becoming too narrow. Diagonal contours can be added to show arrival dates.

for the same transfers. The porkchop plot uses the same contour lines and shows the relatively small region where ballistic transfers are possible. The magenta dots show the maximum mass delivered for each given TOF, which is essentially the ridgeline. This is the same data shown by a single power level (in this case 23 kW) in Figure 3.

There are a few items of note when assessing a bacon plot. The first is that the contours are typically open rather than closed, creating continuous bands of launch opportunities with TOFs fluctuating with the synodic cycle. Rarely are discontinuities found, and when they do appear, it is typically due to optimizer convergence issues. Each point on the plot represents an optimal solution for the date pair, but there are suboptimal solutions below each point. It is common for optimization software to find these "under-families" in certain regions, depending on how the algorithm seeds the subsequent run. For more details on the features and characteristics of bacon plots, see references 1 and 3.

As previously mentioned, each bacon plot must be defined by a number of assumptions. The transfers are not mission independent as they are with porkchop plots. The following parameters define a unique bacon plot:

- Starting state (Starting mass and location/orbit or LV performance curve)
- Ending state (Mars rendezvous, orbit, entry velocity, etc.)
- Thruster set (includes type, mode, quantity of active thrusters)
- Power level (power available to SEP, typically defined at 1 AU)

Note that the trajectory between that starting state and the ending state can be comprised of multiple segments or phases. For example, from launch vehicle to LMO is comprised of a heliocentric leg (launch C3 to Mars rendezvous, $V_\infty = 0$ km/s) and a spiral leg down to low orbit. The combined masses, ΔV's, durations, etc. are represented in bacon plot contours. Typically, either the starting state or ending state will contain the objective function of the optimizer[*], while the other contains an end constraint (e.g. specifying a starting mass and maximizing the ending mass). In theory, a bacon plot could be constructed for the combined phases of an Earth-Mars-Earth round-trip, as long as the parameters for each segment are clearly defined. In complex mission formulation, however, it is usually more efficient to create bacon plots for each leg and survey the possible combinations with a search algorithm.

CREATING AND INTERPRETING BACON PLOTS

The first step to create a bacon plot is to determine the inputs and ranges. These will be dictated by the trade space being explored and the fidelity or granularity desired (which affects total run time). As mentioned, for our work, mission design analysis was carried out using MALTO. Individual trajectories are optimized on the order of one per second, with a wide variety of inputs and outputs. MALTO numerically optimizes the heliocentric transfer leg that starts at Earth escape and ends when it has matched Mars's position and velocity, but does not include Mars gravity. It can, however, append a simple capture spiral to the desired final circular orbit using an analytic

[*] Note that figures-of-merit regarding total transfer time are not considered in bacon plot creation since start and end dates are specified as the plot is created. Time of flight can be considered and optimized later in the reading of the bacon plot during the mission design application. However, if the transfer represented in the bacon plot is polyphasic (e.g. including a spiral at a planet), then some optimization scheme or weighting will need to be placed on the division of total time between the phases.

method.[21] Launch vehicle performance curves, thruster flow rate and thrust vs. power curves, and solar array models are all input as curve fit coefficients. Other considerations such as duty cycle, propellant constraints, forced coast arcs, etc., can also be specified.

Bacon plots can be run over any range, but we have found it most convenient for each span one full Earth/Mars synodic cycle, which rounds to 780 days. The heliocentric transfers can be as short as 150 days, and as long as desired. Typically 800-1000 days captures most of the dynamics and is about as long as most missions would allow. Suggested date ranges for opportunities are given in Table 1. Of course, custom ranges may be used, but these display one full cycle including the region where shorter TOFs are possible. Standard date ranges facilitate easier comparisons to other plots, and can be concatenated to span many opportunities.

Table 1 – Suggested date ranges for single opportunity bacon plots. These are designed to capture the regions of favorable alignment where shorter mass optimal transfers are possible. These regions are many months wide, only approximately centered on the dates given. These opportunities can be used for both Earth-to-Mars and Mars-to-Earth bacon plots. The table can be extrapolated by adding 780 days to each date per opportunity. (JD = Julian Date)

Opportunity	Start Date	End Date	~Optimum*	JD Start	JD End	JD Optimum*
2020	10/8/2018	11/26/2020	6/29/2020	2458400	2459180	2459030
2022	11/26/2020	1/15/2023	8/18/2022	2459180	2459960	2459810
2024	1/15/2023	3/5/2025	10/6/2024	2459960	2460740	2460590
2026	3/5/2025	4/24/2027	11/25/2026	2460740	2461520	2461370
2028	4/24/2027	6/12/2029	1/13/2029	2461520	2462300	2462150
2030	6/12/2029	8/1/2031	3/4/2031	2462300	2463080	2462930

*True "optimum" region can vary by up to +/- 60 days, depending on specifics

The bacon plot can be sampled at any resolution desired, depending on needs and time available. We have found that 10 or 20 day spacing is typically sufficient to capture the salient features and create smooth contours. This requires about 6400 or 1600 trajectory optimizations, respectively. Most of the data is easily interpolatable, with the exception of regions near the lower feasibility boundary. Since each plot is defined by only one power level, and we often wish to optimize power level during formulation, we need to sweep through a range of powers by creating multiple power levels. Again, range and resolution is subjective, but we have found 5-10 levels sufficient for interpolation. The minimum power value should be the level needed to power the engine at the lowest throttle point at the maximum solar distance, while the maximum need not be much higher than the highest throttle point at the same range. Power levels can be equally spaced, or follow a geometric progression such as: 24kW, 26kW, 30kW, 35kW, etc.

Every optimizer has its quirks and sensitivities, and MALTO is no different. Finding robust convergence over a wide range of parametric sweeps is finicky at best. As with other tools, there are many knobs to turn to fix problems. The location seed case in the LD/AD space can be very important, as is the direction in which the space is sampled, especially if the previous solution is being used for the subsequent seed. Figure 5 gives an example of a particularly difficult case. The initial parametric sweep, especially when trying to cover a large date range, can often fail or give spurious results. Sometimes the optimizer will converge on solutions that are from a family of transfers that are not the true optimum for the region of the bacon plot. These often become apparent after making the contour plot and seeing abrupt contour shifts (see Figure 5a and 5b). In order to get better convergence for the whole range, and to eliminate the "underfamilies", it is sometimes necessary to break the plot into smaller regions and to start with multiple seeds. It can also help to sweep through the parameters from varying directions (e.g. horizontal/vertical, left/right, see Figure 5c). Even after a few passes through the full date range, sometimes point or regions remain unconverged or suboptimal, but these can be removed with smoothing or interpolation.

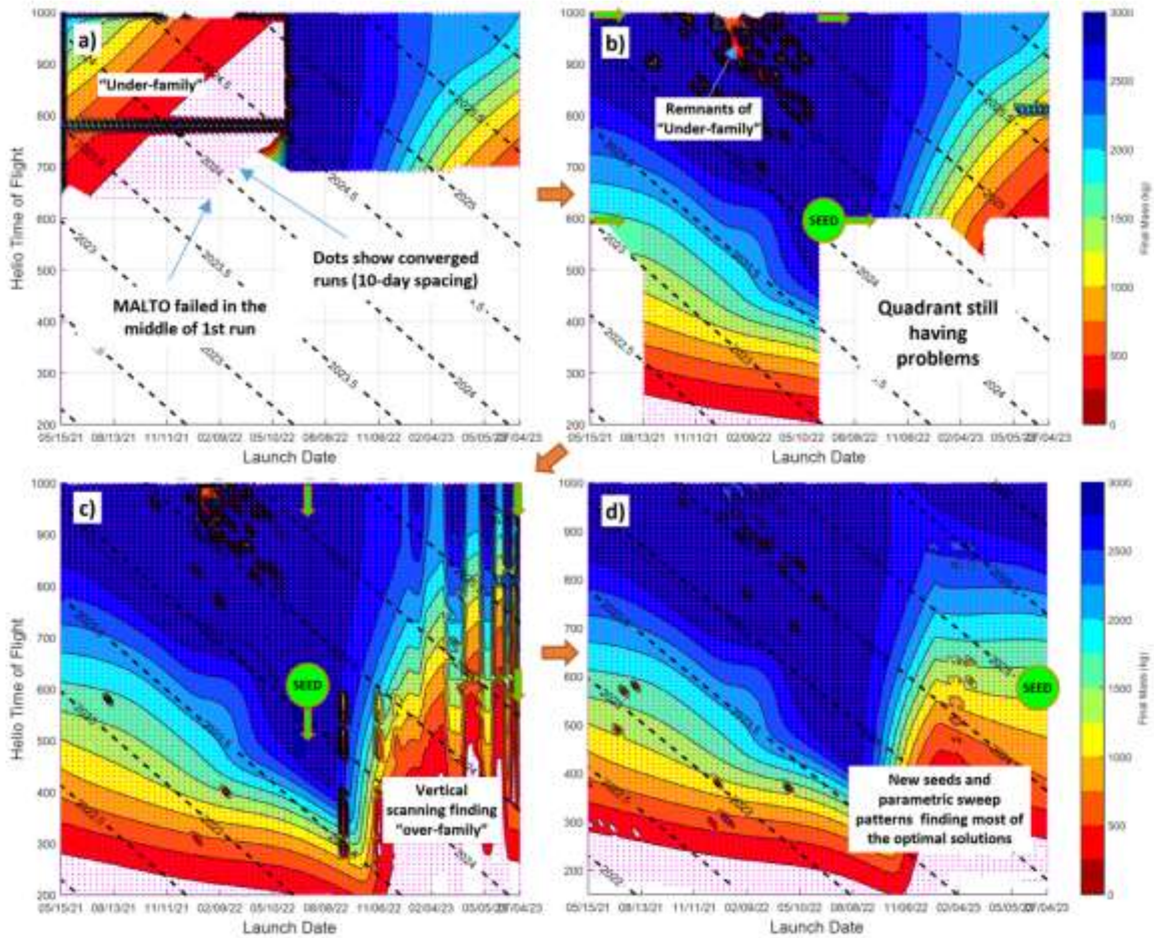

Figure 5 – Intermediate steps in creating a bacon plot showing possible convergence issues. Some optimization methods and trajectory types are more robust than others. In this example: a) the parametric sweep of LD and TOF continued horizontally from the upper left, finding a sub-optimal family and failing early, b) the run was repeated by sweeping four quadrants horizontally, c) the quadrants were then repeated vertically, from right to left, and d) various patterns were used to find the optimal transfers for each date pair.

ADVANCED APPLICATION

Once all of the parameters have been decided, and a "pile" of bacon plots has been created, we can then use them for overall system optimization for a specified mission. In the most basic application, the contours on the plot give the amount of propellant needed to deliver a specified mass in a specified amount of time. Simple estimating relationships can then be used to size the propulsion system, which changes the mass, which changes the dates, etc., and the process is repeated to convergence. Unfortunately, this simple method does not take into account all of the other various parameters and choices at the system and mission levels that affect total mission feasibility and optimization.

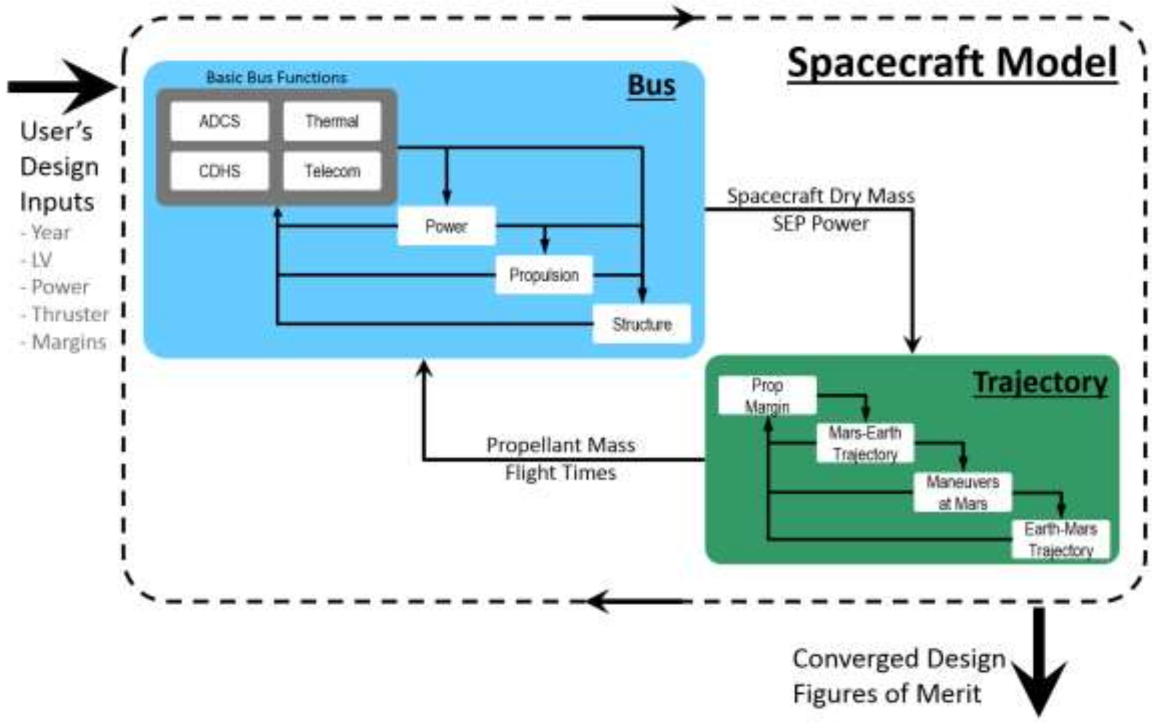

Figure 6 - Process flow for spacecraft mission design tool. Not only are the spacecraft subsystems interrelated, but the SEP trajectories themselves are part of the iteration loops.

At JPL, we have developed techniques and tools to read and interpolate bacon plots in the design of highly complex round-trip missions to Mars.[22] Detailed sub-system models are all interconnected with the inputs and outputs of both Earth-Mars and Mars-Earth bacon plots. This method allows us to sweep through giant option spaces in search of optimized mission architectures that meet mission constraints. In support of this effort, hundreds of millions of trajectories were optimized to create a vast database for the tool to traverse. The sophisticated design tool was then able to query and interpolate the database as it sought to meet the numerous constraints of a potential Mars sample return campaign. Promising mission architectures were discovered that creatively met the challenging mass and timeline constraints that previously would have been undiscovered without the data from the bacon plots.

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REFERENCES

[1] R. Potter, R.C. Woolley, A.K. Nicholas, and J. Longuski, "Features and Characteristics of Earth-Mars Bacon Plots," *AIAA/AAS Astrodynamics Specialist Conference*, Stevenson, WA, Aug. 2017.

[2] G. Genta and P.F. Maffione, "Interplanetary Missions Performed Outside the Optimal Launch Windows," IAC-17-A5.2, *68th International Astronautical Congress (IAC)*, Adelaide, Australia, 25-29 September 2017.

[3] R.C. Woolley and A.K. Nicholas, "SEP Mission Design Space for Mars Orbiters," *AIAA/AAS Astrodynamics Specialist Conference*, AAS Paper 15-632, Vail, CO, Aug. 2015.

[4] R.C. Woolley, J.D. Baker, D.F. Landau, and K.E Post, "Low-Thrust Trajectory Maps (Bacon Plots) to Support a Human Mars Surface Expedition," *AIAA/AAS Astrodynamics Specialist Conference*, AAS Paper 17-652, Stevenson, WA, Aug. 2017.

[5] R.R. Stephenson. "Electric Propulsion Development and Application in the United States," Proceedings of the 24th International Electric Propulsion Conference (IEPC-95-01), Moscow, Russia, 1995.

[6] D. Lev, R. Myers, et. al., "The Technological and Commercial Expansion of Electric Propulsion in the Past 24 Years." *35th International Electric Propulsion Conference*, Atlanta, GA, Oct. 8 – 12, 2017.

[7] M.D. Rayman, P.A. Chadbourne, J.S. Culwell, and S.N. Williams, "Mission Design for Deep Space 1: A Low-Thrust Technology Validation Mission," *Acta Astronautica*, Vol. 45, Issues 4–9, August–November 1999.

[8] H. Kuninaka, J. Kawaguchi, "Lessons Learned from the Round Trip of Hayabusa Asteroid Explorer in Deep Space" IEEE Aerospace Conference, Paper #1771, 2011.

[9] Y. Tsuda, M. Yoshikawa, M. Abe, H. Minamino, S. Nakazawa, "System design of the Hayabusa 2—asteroid sample return mission to 1999 JU3." *Acta Astronautica*, Vol. 91, pp. 356–362, 2013.

[10] G. D. Racca, A. Marini, L. Stagnaro, J. Van Dooren, L. Di Napoli, B. H. Foing, R. Lumb et al. "SMART-1 mission description and development status." *Planetary and space science* Vol. 50, no. 14-15, pp. 1323-1337, 2002.

[11] C. T. Russell, M. A. Barucci, R. P. Binzel, M. T. Capria, U. Christensen, A. Coradini, M. C. De Sanctis et al. "Exploring the asteroid belt with ion propulsion: Dawn mission history, status and plans." *Advances in Space Research* Vol. 40, no. 2, pp. 193-201, 2007.

[12] D.G. Yarnoz, J. Ruediger and P. De Pascale, "Trajectory design for the Bepi-Colombo mission to Mercury." *Journal-British Interplanetary Society*, Vol. 60, no. 6, p.202, 2007.

[13] K. deGrys, A. Mathers, B. Welander, and V. Khayms, "Demonstration of 10,400 Hours of Operation on a 4.5 kW Qualification Model Hall Thruster", AIAA Paper 2010-6698. Presented at the 46th Joint Propulsion Conference (JPC), Nashville, TN, 2010.

[14] J.S. Snyder and R.R. Hofer. "Throttled Performance of the SPT-140 Hall Thruster." Presented in the 50th AIAA/ASME/SAE/ASEE Joint Propulsion Conference, 2014.

[15] V. Vial, J. Vaudolon, O. Duchemin, N. Cornu, and J.M. Lonchard, "Electric Propulsion at Safran," IAC-17,C4,4,x41304, presented at the 68th International Astronautical Congress, Adelaide, Australia, 2017.

[16] S.D. Clark, M.S. Hutchins, I. Rudwan, N.C. Wallace, and J. Palencia. "BepiColombo Electric Propulsion Thruster and High Power 1 Electronics Coupling Test Performances," Proceedings of the 33rd International Electric Propulsion Conference (IEPC-2013-133), Washington D.C., 2013.

[17] R. Hofer and H. Kamhawi, "Development Status of the 12.5 kW Hall Effect Rocket with Magnetic Shielding (HERMeS)," Presented at the 35th International Electric Propulsion Conference (IEPC), IEPC-2017-231, Atlanta, GA, Oct. 8 - 12, 2017.

[18] J. A. Sims, P. A. Finlayson, E. A. Rinderle, M.A. Vavrina, and T. D. Kowalkowski, "Implementation of a Low-Thrust Trajectory Optimization Algorithm for Preliminary Design," AIAA/AAS Astrodynamics Specialist Conference and Exhibit, Keystone, Colorado, Aug. 2006.

[19] J. Englander, "Rapid Preliminary Design of Interplanetary Trajectories Using the Evolutionary Mission Trajectory Generator," 6th International Conference on Astrodynamics Tools and Techniques (ICATT), Darmstadt, Germany, Mar. 14-17, 2016.

[20] M. Vasile and R. Jehn, "DITAN: A Tool for Optimal Space Trajectory Design," https://trajectory.estec.esa.int/Astro/3rd-astro-workshop-presentations/DITAN-%20A%20Tool%20for%20Optimal%20Space%20Trajectory%20Design.pdf

[21] Melbourne, W. G. and Sauer, C. G., "Performance Computations with Pieced Solutions of Planetocentric and Heliocentric Trajectories for Low-Thrust Missions," Space Programs Summary 37-36, vol. IV, Jet Propulsion Laboratory, Pasadena, California, pp. 14–19, December 31, 1965.

[22] A.K. Nicholas, R.C. Woolley, A. Didion, F. Laipert, Z. Olikara, R. Webb, and R. Lock, "Simultaneous Optimization of Spacecraft and Trajectory Design for Interplanetary Missions Utilizing Solar Electric Propulsion," AAS Astrodynamics Specialist Conference. Maui, HI, Jan. 2019.